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PULSED PLASMA PROPULSION TECHNOLOGY

FINAL REPORT FOR PERIOD 10 MARCH 1972 - 30 MAY 1977

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FOREWORD

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A pulsed plasma thruster system delivering one millipound of thrust at a sustained firing rate of .16 Hz and specific impulse of 1620 seconds was developed, designed, fabricated, and tested for a total of 12,183 pound-seconds (55,714 Newton-seconds) of total impulse prior to voluntary shut-down of the test. The system utilizes high energy density 40 Joules/pound (81.7J/kg) KF-film capacitors and a completely packaged bread-board power converter. A total impulse capability of 37,500 ———

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pounds-seconds (166,810 Newton-seconds) was designed into the system and the resulting 23.44 pounds (10.63 kg) of solid Teflon propellant is stored in the shape of a helical coil having an outside diameter of 17.32 inches (44 cm) and overall height of 6.25 inches (15.88 cm). The system was refurbished after completion of the endurance test so that the total compliment of propellant could be restored and run to complete depletion at the AFRPL during their in-house endurance test. This report describes in detail the system and results of endurance testing at Fairchild Republic Company.

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ILLUSTRATIONS

1.0 INTRODUCTION

Air Force requirements for low thrust propulsion systems capable of delivering a large total impulse for use on satellites to provide auxiliary control functions have inspired serious consideration of electric thrusters because of their high specific impulse. At the onset of the program reported upon herein, pulsed plasma thrusters using solid Teflon * propellant had recently been scaled up from micropound thrust levels to the millipound thrust level required by the Air Force. Those initial scaling efforts had been carried out under NASA Langley R. C. funding.[1,2] The first phase of this program was initiated with the sole objective of optimizing performance of a millipound thruster. In particular, a goal of 30% thrust efficiency at 1500s specific impulse was established by the AFRPL for that phase. This objective was met and demonstrated several times during the program[3]. Having met the performance requirement, a second phase effort was initiated in which a laboratory thruster system capable of delivering 37,500 pound seconds (166,810 Ns) complete with breadboard electronics was to be designed, fabricated, and endurance tested up to 10,000 pound seconds (44,480 Ns) at 100% duty cycle. This second phase effort required considerable increase in the state-of-art capacitor energy density, as well as evaluation of several new concepts regarding storage of the required 25 pounds (11.34 kg) of propellant and feeding of the propellant into the newly developed side-fed nozzle configuration. The test thruster system was fabricated using helical teflon propellant coils to minimize the volume of the package and capacitors manufactured by Capacitor Specialists, Inc. in which KF-film (polyvinyl fluoride) dielectric was utilized. Concurrent with the fabrication of the helical rod system, a test thruster using straight propellant rods was also assembled so that the concept of feeding propellant into the nozzle from the sides could be verified. The initial helical coil propellant system was never run for any significant period of time because of problems with the feed system, strip line insulation breakdowns, and capacitor leakage. The complete results of the initial work done on the system were reported upon in Reference 4.

Although the early attempts at fabricating the millipound system to meet the endurance requirements established as a goal for this program were not successful from the standpoint of meeting that goal, considerable knowledge was gained as a result of those efforts. In particular, three major problem areas were identified for which reasonable solutions were formulated. These problems were

    (i)   Capacitor leakage in a vacuum environment due to poor can
            construction and insufficient Quality Assurance during manufacture.

* DuPont's trade name for Polytetrafluoroethylene

(ii)   Inadequate thruster strip line insulation in various "trouble spots".

(iii) Propellant rod binding resulting from large frictional forces between propellant rod sections and a feed system which was not truly independent for each section of propellant.

The solutions proposed to eliminate those problems were:

(i)   Procure new capacitors with the can design and manufacture controlled by Fairchild Republic Company. Utilize tougher screening specs during each phase of production.

(ii) Completely redesign the insulation of the capacitor strip line to accept the new capacitors and eliminate local electrical breakdown problems.

(iii) Modify the existing feed system so that individual propellant rod sections could be fed in a more independent fashion and minimize friction between rods.

Two Interim Reports were released during this effort. The first, AFRPL-TR-73-79; contains information relevant to attaining the desired thruster performance and the second, AFRPL-TR-74-50; describes the basic original design and operation of the thruster system in addition to preliminary data on radiated and conducted EMI.

The work performed to incorporate the above solutions and the resulting successful generation of 12,183 pound seconds (54,192Ns) of total impulse before voluntarily discontinuing the endurance test forms the basis for the majority of that portion of this program reported upon herein. In addition to the above, some new problem areas were uncovered which surfaces only as a result of being able to run the propulsion system for a long period of time in a vacuum. These problems and the steps taken to resolve them are also discussed in this report.

2.0 GENERAL DESCRIPTION OF MILLIPOUND THRUSTER SYSTEM

Solid propellant pulsed plasma propulsion systems may generally be broken down into six major component subassemblies. These are the power converter, energy storage capacitors, strip line and electrode assembly, discharge initiating (DI) circuit, propellant, and supporting structure. Each of these components is described below.

2.1 Power Converter

The function of the power converter is to transform the input DC power typically found on spacecraft into energy stored in the capacitor bank at a predetermined rate, dependent upon the firing frequency desired. This is accomplished by charging the capacitors to the appropriate voltage necessary to generate the required energy. In the millipound thruster system, a total capacitance of 275 microfarads is charged to 2335V for a total of 750J. Charging occurs in a maximum of 5.38 seconds, yielding an average output power of 139.4W in that period of time. The input voltage to the converter is 28 ± 2 VDC under load.

In addition to the above output, the converter also provides a charge of 620V to the 2 microfarads of capacitance in the DI circuit. The overall efficiency of the packaged breadboard coverter manufactured by Wilmore Electronics, Inc., is 80%. A schematic and listing of circuit components can be found in Reference 4. It is noted that although the converter procured for this program is a breadboard version, the dimensions of the exterior case are only 2.5 x 8 x 10 inches. (6.35 x 20.32 x 25.4cm) and the weight is 5 pounds (2.27 kg).

2.2 Energy Storage Capacitors

Energy storage capacitors were specially manufactured for this program to ensure a reliable hermetic seal. Windings, impregnation, and final sealing were done by Maxwell Laboratories, Inc. whereas the actual cans were provided by Fairchild Republic Company which supervised the construction and provided the necessary quality assurance specifications. Details concerning the manufacture of these capacitors will be found in Reference 5. The work was funded by the NASA with Jet Propulsion Laboratories acting as contract administrators.

Four capacitors connected in parallel by a low inductance strip line assembly are utilized on the millipound thruster. Each capacitor has a capacitance of approximately 68 microfarads and weighs 5 pounds (2.27 kg) for an energy density of 37J/pound (81.7J/kg) at 2335V. The dielectric system is composed of KF-film, paper, and castor oil. These units have been derated from 5KV and as such have a life expectancy in excess of ten million discharges at the thruster operating voltage. The capacitors are cylindrical, having a diameter of 4 inches (10.16cm) and length of 6 inches (15.24cm).

2.3 Discharge Initiating Circuit

A schematic of the DI circuit can be found in Reference 4. This circuit is used to fire the solid state ignitor plug which is located in a ceramic housing just above the thruster cathode. The plug cathode and thruster cathode are electrically separated by one Ohm of resistance. The DI circuit produces a fast rising voltage pulse across the plug causing a sliding discharge on the surface of the exposed semiconductor which separates anode from cathode. This microdischarge extracts a miniscule amount of semiconductor surface material which is accelerated across the interelectrode gap through a .25 inch (6.4mm) diameter hole in the thruster cathode causing breakdown to occur between the thruster cathode and thruster anode.

2.4 Teflon Propellant

A thruster specific impulse of 1600s and design total impulse of 37,500 pound

seconds (166.810NS) lead to a requirement for 23.44 pounds (10.63 kg) of propellant. In order to minimize the overall system volume, the propellant is stored in the shape of a helix, having an inside diameter of 14.70 inches (.373m) and outside diameter of 17.32 inches (.440m). The cross section of the propellant rods is rectangular, having a height of 3.0 inches (7.62cm) and width of 1.31 inches (3.33cm). The helical pitch is 3.25 inches/360°. Two separate helical rods are fed into the interelectrode gap from either side. The rods are subdivided into three separate sections to form the 1.31 inch overall width. The innermost section is .66 inches (1.68cm) wide while the middle and outside sections are both .33 inches (.84cm) wide. Each of the sections are fed into the interelectrode gap using an independent spring feed system. The inner section was fed using a spring loaded motor pulling a cable attached to the rear end of the rod and the other two sections were fed using negator constant force springs. It is noted that although the innermost rod measures .66 inches wide, it was initially manufactured as two individual rods each measuring .33 inches wide. These were later bonded together by etching the surfaces to be bonded with a sodium solution and then cementing the etched rods together with a fluorocarbon cement. The reason this was done is explained in Reference 4. In all future programs, the innermost rod will be manufactured as a single piece but for purposes of the endurance test reported upon herein, the cemented innermost rod was used.

The helical propellant rods are manufactured by first having individual cylinders molded to approximately the correct ID and OD. These are then machined to the correct diameters such that they fit snuggly, one within the other, similar to a telescope. Each set of three cylinders is then placed on a tracing machine which follows the appropriate helical contour to produce the rods. A photograph of the cylinders during the machining process is presented in Figure 1.
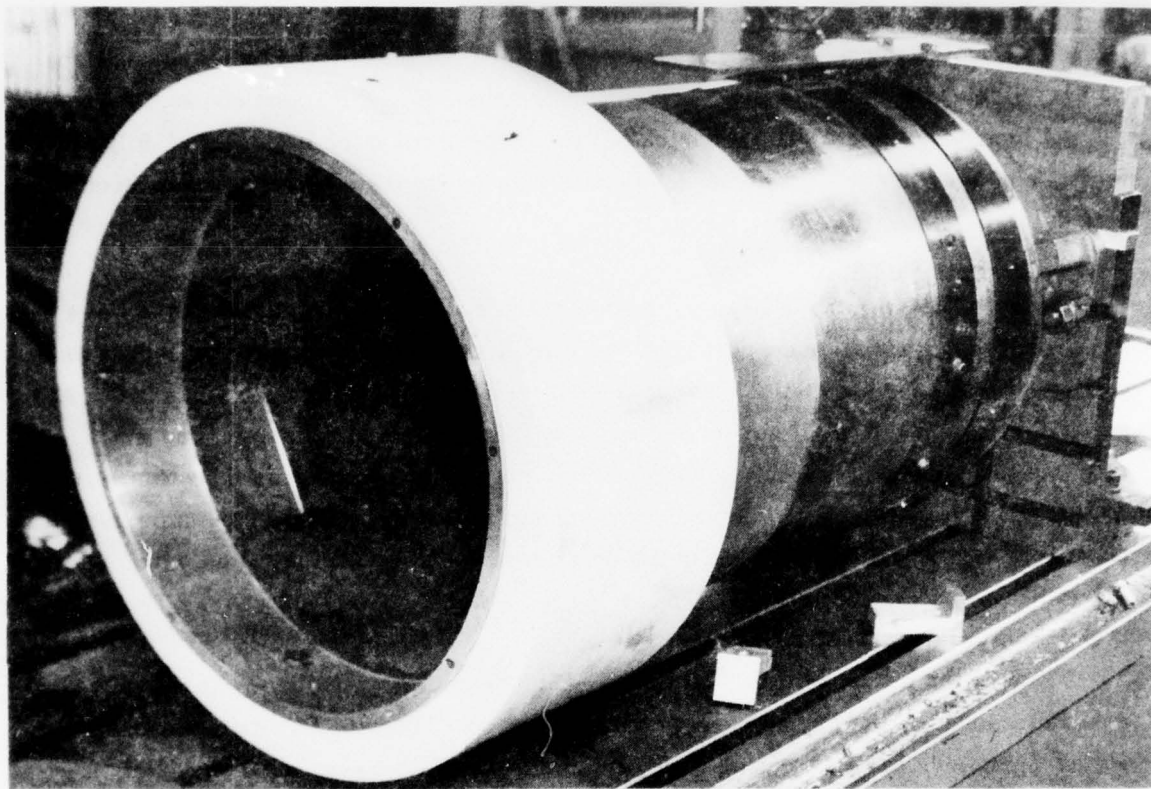
Figure 1.   Teflon Cylinder During Helical Machining Operation

2.5   Strip Line and Electrode Assembly.

The four capacitors are connected electrically in parallel through a low inductance strip line assembly.   The strip lines are fabricated using .032 inch (.83mm) copper sheet.   Mylar is used between the strip lines for electrical insulation and the entire exterior surface, including edges, is insulated using .032 inch thick NEMA G-10 phenolic.   The phenolic exterior insulation also provides structural rigidity to the strip lines.   Electrode extension components are fastened to the center of the strip line assembly.   These are fabricated using .062 (1.6mm) copper and are also insulated using G-10.   The surface of the extensions fore of the nozzle is utilized

to intercept heat generated at the electrode before it can conduct back to the capacitors. Electrical isolation of this parallel heat conduction path is provided by placing a slab of Beryllium Oxide between the surface of the extensions and the heat conductor line.

Electrodes were initially fabricated from copper but later in the test, Molybdenum and Tungsten were also used. Details concerning the electrodes are presented in a subsequent section of this report.

2.6 Structural Support.

The basic approach used in the design of the supporting structure consists of a centrally located circular support ring which is the main member and also serves as an alignment plane for assembly. The power converter, propellant support brackets, and capacitor assembly are all mounted directly to this member. It is noted that since this thruster was intended only to be a test system having no shock or vibration qualification requirements, the structural support was limited to the bare minimum necessary to support the thruster in the horizontal mounting configuration only.

3.0 PRELIMINARY TEST RESULTS

3.1 Intermediate Thruster System

During the period of time that the KF-film capacitors were being manufactured (see Reference 5), the thruster was modified to accept Maxwell Lab's off-the-shelf Mylar capacitors having a capacitance of 60 microfarads. These units were, of course, much larger both dimensionally and weight-wise than the KF-film capacitors were expected to be. The 60 microfarad off-the-shelf Maxwell units weigh 12 pounds (5.44 kg), are rated at 8KV, and measure 6.5 inches (16.50cm) long by 6 inches (15.24cn) in diameter. These capacitors were adapted to the existing strip line assembly using four elbow-shaped strip line assemblies specially designed and fabricated for that purpose. A photograph of the thruster system with the four Mylar capacitors is presented in Figure 2.
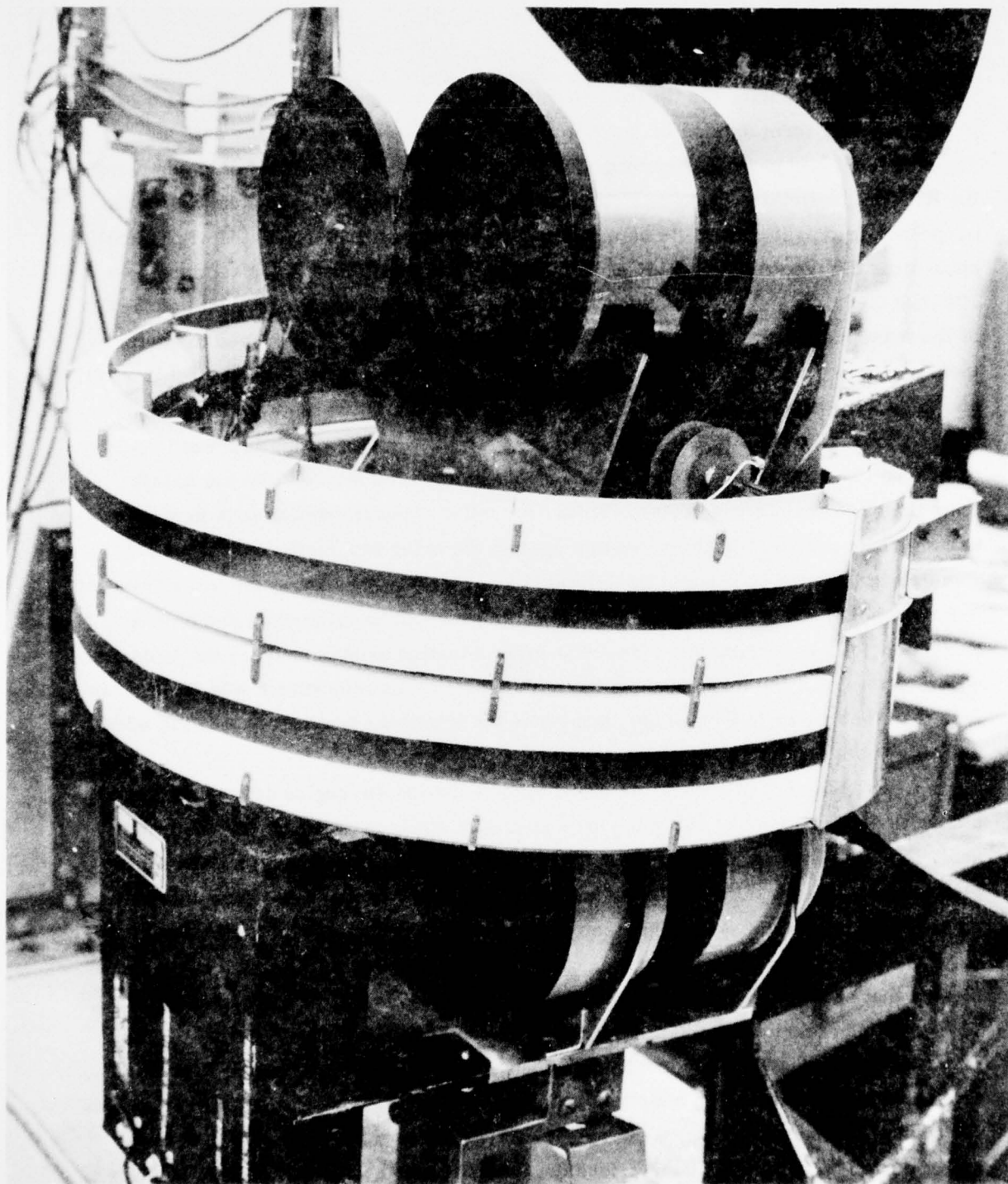
Figure 2. Photograph of Intermediate Thruster System With Mylar Capacitors

The purpose for running the thruster in this intermediate configuration was to begin generating long term data on the system as soon as possible so that any deficiencies could be identified prior to beginning the actual endurance test which would run using the KF-film capacitors being manufactured. In particular, new modifications to the propellant feed system were to be evaluated. These inovations consisted of physically separating the innermost propellant section from the middle and outer sections by a .04 inch (1.0mm) strip of aluminum mounted rigidly to the support brackets at the OD of the inner rod section, and using an independent separate spring to feed the outermost section. The .04 inch aluminum strip (referred to henceforth as the splitter rail) and the spring are evident in Figure 2. The purpose for the splitter rail, which penetrates about .06 inches (1.5mm) between the inner and middle propellant sections, is to take up the outward radial load induced by pushing the inner rod into the nozzle from the back end. It had been found during earlier experimentation (see Reference 4) that this outward force pushing directly against the outer two sections of propellant was causing binding because of the friction incurred. By separating the inner from the outer sections, the frictional force on all sections would be minimized and binding would hopefully be eliminated. Much the same situation existed between the middle and outer sections but the force pushing the middle section into the nozzle was considerably less than that on the inner section, and hence the frictional loading between the middle and outermost sections was considered to be a lesser factor.

After redundantly sealing the capacitors by potting the region around the center stud with semirigid epoxy and epoxying a specially machined ring around the rear lid seal, the thruster was placed on test. Several problem areas became apparent as testing went on. First, it became obvious that certain regions of the strip line assembly, particularly in the vicinity of the electrode extension attachment point, needed greater insulation resistance. Further attempts at better insulating this area led to the conclusion that a redesign of this region of the strip lines would have to be accomplished. Thus, prior to fabricating the strip lines which would be used with the new KF-film capacitors, a redesign of the extension attachment area was performed. A second problem which arose during testing of the intermediate thruster was breakdown across the edge of the Beryllium Oxide electrical insulation/thermal conduction interface slab mounted on the high voltage extension. This interface was also redesigned as a result of the problems encountered during preliminary testing. The final problem which surfaced during these initial tests was associated with the power converter. As the temperature increased during thruster firing, the energy delivery rate (output power)

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would decrease until thermal equilibrium was reached. Since the temperature at
equilibrium was within the range specified to Wilmore Electronics, Inc. and the
energy delivery rate at thermal equilibrium was considerably less than the minimum
specified, it was apparent that the heat dissipative components within the converter
were not properly heat sunk to the conduction path leading to the mounting flanges of
the converter. This fact became further evident when a power transistor in one of
the four output channel breadboards failed because it overheated. Upon discussing
the problem with Wilmore Electronics, the decision was made to run the thruster
with the converter outside the vacuum chamber. This decision was made because
time and funding limitations made sending the converter back to Wilmore for modi-
fication prohibitive. This design problem was, however, solved later in the program
(see Section 6.0).

All of the problems cited above were considered manageable from an engineering
standpoint and represented no real threat to the eventual propulsive or life capability
of the system. The last problem encountered, however, did pose a more fundamental
question. That problem was electrode erosion, or, more specifically, erosion of the
anode which aroused attention. The reason for concern stems from the fact that the
only surface of the anode which is exposed to the discharge within the propellant gap
is the propellant retaining shoulder. If erosion of this shoulder were severe enough
to completely eliminate its ability to hold any one of the spring loaded propellant rods
in position, then that rod would move across the gap and close off the nozzle. This,
therefore, would represent a possible failure mode for the system. It is noted that
if there were no need for a propellant retaining shoulder on the anode surface, one
would not consider anode erosion to represent a failure mode. Enough anode material
could easily be provided in the discharge region to accommodate whatever total impulse
was required by the system. The need for a propellant retaining shoulder with the
current system design prompted an immediate redesign of the anode and consideration
of materials other than copper. Based on the pattern of the erosion, certain bene-
ficial modifications to the existing electrode design became apparent and a new design
was arrived at. The new anode designed for the endurance test also included a modi-
fication to the surface in contact with the propellant which heretofore had been flat
and was changed to match the helical twist which actually exists on this surface.
Over the length of the anode, this twist is only about 0.6 degrees but it was found that
not taking this small departure from truly flat into account led to undesirable burning
of the propellant rod's undersurface.

3.2 Endurance Test Thruster System

Upon obtaining the new KF-film capacitors and making the modifications to the system found necessary during preliminary testing of the intermediate system, some preliminary testing of the thruster with these capacitors was performed prior to putting the system on test at 100% duty cycle. In particular, thruster performance was verified and a series of thermal-vacuum tests were performed until the appropriate heat balance could be assured.

The approach chosen for maintaining proper thruster operating temperature typifies what will be found on most satellites. Namely, the thruster is firmly bolted to a mounting platform which is part of the satellite structure and which is maintained at a relatively constant temperature by the satellite thermal control system. This platform represents a heat sink to which all heat dissipated by the various thruster components can be conducted. For test purposes, the thruster was mounted to a plate whose temperature was maintained constant by running cool water through a coil in contact with the plate. A second plate with a separate coil was connected in series with the first and fastened to the top of the thruster to simulate the surrounding case normally found on flight hardware. As explained previously, although the power converter was utilized throughout the endurance test, it was not part of the thruster hardware inside the vacuum chamber and hence no thermal load due to converter dissipation (approximately 34W) was experienced by the heat sink. It is noted that later on in this program, the power converter was successfully integrated to operate the thruster system in vacuum prior to delivery of the system to the AFRPL (see Section 6.0)

During one of the iterations to the thermal design performed prior to achieving the proper configuration, capacitor case temperature near the front end was allowed to rise close to the maximum 40°C recommended operating temperature. A capacitor failure occurred shortly thereafter with only 16,577 pulses on the failed unit. It is difficult to determine whether this failure occurred because of the temperature or it was simply the result of what is commonly referred to as "infant mortality". Normally, one of the screening tests performed on flight qualification capacitors is to run the units under most severe conditions with the thruster system for a minimum of 50,000 firings to circumvent the low probability of statistical early failures to which this term is applied.

The results of thermal vacuum testing are presented in Figure 3 where the temperature at various locations on the thruster is plotted as a function of time. The data of Figure 3 were obtained at a pulse rate of one firing every 6.8 seconds for a

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HEAT SINK (CONTROLLED)
ANODE EXTENSION
CATHODE EXTENSION
LOWER RIGHT HAND CAPACITOR
UPPER LEFT HAND CAPACITOR

INPUT POWER TO CAPACITOR BANK = 110.0W
POWER CONVERTER OUTSIDE OF CHAMBER
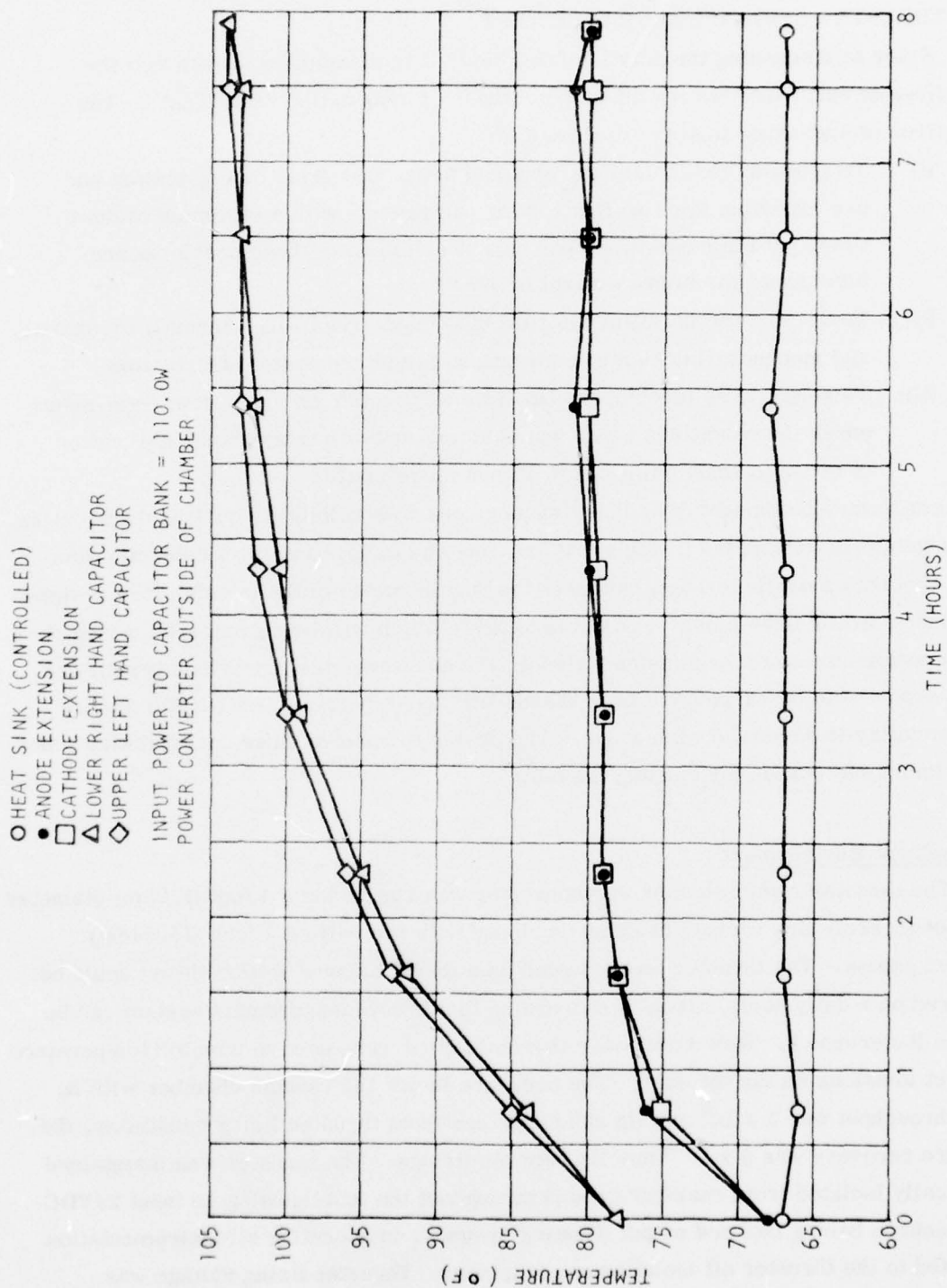
TEMPERATURE (°F)

TIME (HOURS)

Figure 3. Results of Thermal/Vacuum Testing on Endurance Test Thruster System

net power output of 110W to the capacitor bank. Having assured the proper capacitor case temperature under normal operating conditions, the endurance test was initiated. Results of that test are presented in the next section of this report.

4.0 THRUSTER SYSTEM ENDURANCE TEST

Prior to discussing the details of this test, it is meaningful to note that the objectives of endurance testing differ from those of a so-called "Life Test". The objectives of endurance testing are threefold:

(i) To pinpoint and isolate any problem areas that arise during testing and use expedient fixes so that testing can resume with a minimum of down time. At the same time, permanent solutions to these problems are formulated for future system fabrication.

(ii) To develop the necessary engineering inputs regarding thermal, electrical, and mechanical design considerations for future system fabrication.

(iii) To generate as much operating time as possible on all system components under the conditions which would be encountered in an actual application in order to assess the status of system reliability.

Life testing may be thought of as being synonymous with reliability qualification testing. One is generally interested in achieving only one objective; namely, a determination as to whether or not the system can meet the requirements of the mission it is designed for under conditions as close as possible to those which will be encountered during the actual performance of that mission. Hence, the endurance test described herein was never expected to be carried out on a "hands-off" basis because of a certain degree of uncertainty in several design areas. The desire to resolve those uncertainties was the primary motivation for running the test.

4.1 Ambient Environment

The thruster system less power converter was run inside a 4 foot (1.22m) diameter by 6 foot (1.83m) long vacuum chamber equipped with two baffled 6 inch (15.24cm) diffusion pumps. The thruster was mounted on a thrust balance so that thrust could be measured on a daily basis. Details concerning the thrust measurement system can be found in Reference 3. Iron-Constantan thermocouples were used to monitor temperature at select locations on the thruster. The pressure inside the vacuum chamber with no mass throughput was 3×10^6 mm Hg and under continual thruster firing conditions, the pressure recovery was 5×10^{-5} mm Hg between firings. The thruster was maintained electrically isolated from chamber ground throughout the test by using an input 28 VDC power source having isolated output (floating ground) and operating all instrumentation connected to the thruster off isolation transformers. Thruster firing voltage was monitored using a precalibrated 1000:1 voltage divider connected to a 0-4V digital peak reading voltmeter.

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4.2 Thruster Performance Throughout Endurance Testing

The endurance test was not run continuously from start to finish due to a number of interruptions. These were the result of facilities malfunction, voluntary shut downs for visual inspection, and thruster operating problems. All shut downs are cited and explained in section 4.3. A total of 12,183 pound seconds (54,193Ns) of impulse were accumulated during the period of time from 28 January 1976 to 4 October 1976. During that time, the system had been running at 100% duty cycle for 4,500 hours of the total 6,650 hours in that time period. Impulse bit measurements were made daily except on weekends and total impulse accumulation was computed by numerically integrating the impulse bit data from day to day. A summary of the data is presented in Figure 4 where impulse bit and total impulse are plotted as a function of the number of discharges. Observe that after approximately 76,000 discharges, the impulse bit magnitude is remarkably constant right up to the voluntary termination of the test after 2,445,785 discharges had been accumulated. The average impulse bit (subsequent to the initial 76,000 discharges) obtained by dividing the total impulse accumulated by the total number of discharges turns out to be 4.961 mlb-s (22.069 mN-s) and the maximum deviation of the measured impulse bits from this average is within the range 4.562-5.348 (i.e. 4.961 ± 8%). The initial drop-off of the impulse bit from a high of 6.568 mlb-s (29.216mN-s) to 5.215 mlb-s (23,197mN-s) after 76,000 firings is attributable to the propellant rods "burning in" to the equilibrium surface contour which exists thereafter and being held in that position by the fuel retaining shoulder. The impulse bit existing after burn in is more than likely lower in magnitude than that which would exist had this test been run with innermost propellant sections which were not previously cemented together (see section 2.4). This conjecture is made on the basis of the observed unusual manner in which the innermost sections were seen to burn. A carbonized layer between the two cemented Teflon pieces is etched into the surfaces, enabling the two to be stuck together. This carbonaceous layer was apparently acting like a heat shield to the Teflon beneath, not allowing the surface to sublime as it normally would. As a result there was a discontinuity in the ablation surface of the innermost section along the glue line which persisted throughout testing. In all probability, had this propellant section been fabricated as a single piece (which it was on the thruster used for plume contamination studies[6]) additional mass and a more uniform discharge channel would have been obtained and the impulse bit would therefore have been larger. For example, the impulse bit amplitude remained at 5.98 ± 3% millipound-sec for the duration of the plume contamination study with the identical electrode nozzle contour.
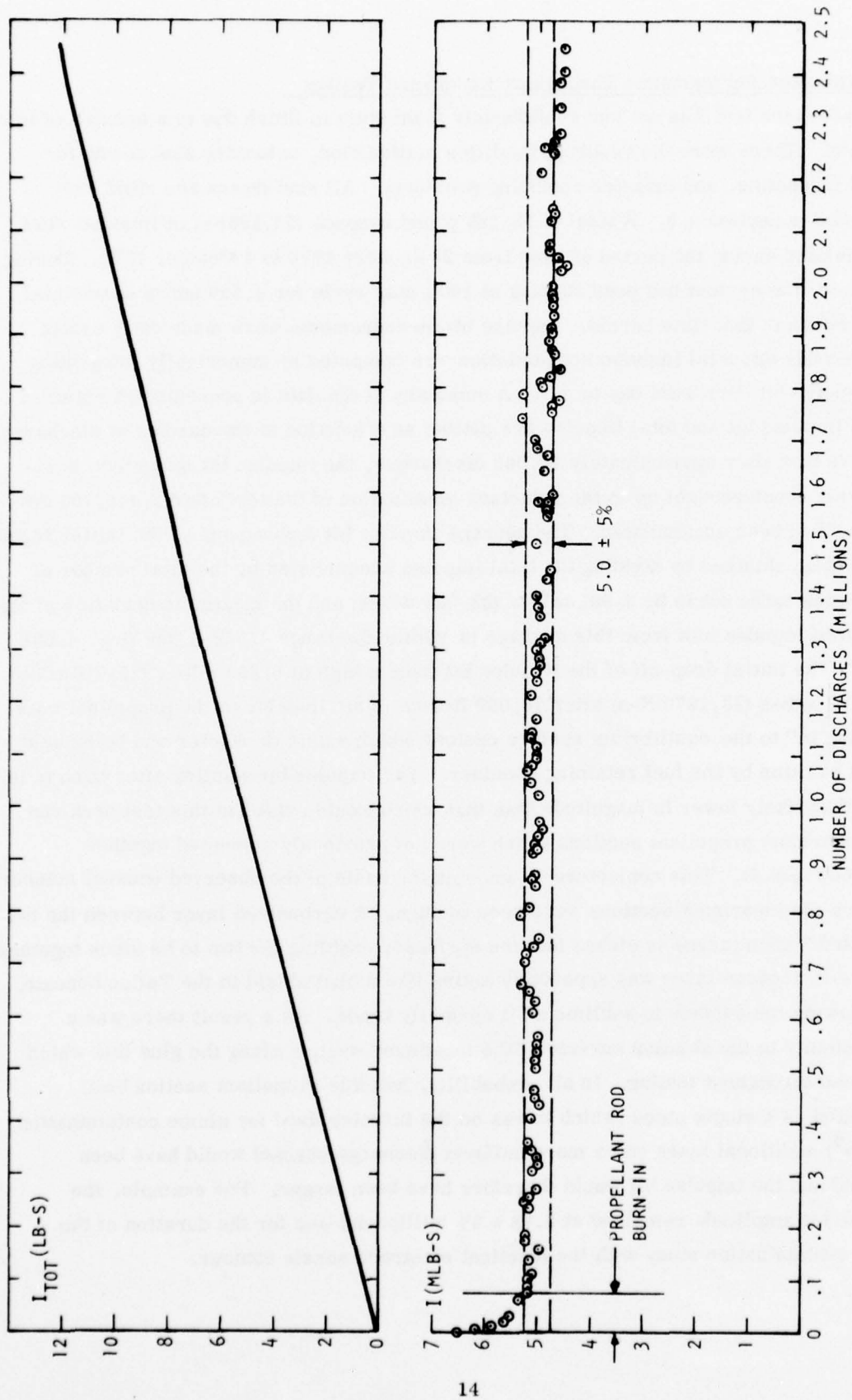
Figure 4. Impulse Bit and Total Impulse Vs. Number of Accumulated Discharges

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Although specific impulse could not be monitored as a function of time during the test without having to invest a considerable amount of time, a measurement of the I_{sp} was made towards the tail end of the test. The I_{sp} (Total impulse/propellant weight used) measured over the last 1100 pound seconds of operation was 1626 seconds.

4.3 Chronology of Events

The chronology presented in Table I is a summation of the test history. All shut-downs are indicated numerically in chronological order and the reason for a shutdown is categorized as one of the following:

 (i) Voluntary shutdown for visual inspection of thruster.

 (ii) Facility malfunction requiring shutdown of thruster.

 (iii) Mandatory shutdown due to thruster malfunction.

The following text describes the conditions and actions taken during each shutdown in the numerical sequence indicated in Table I.

1. On Sunday, 8 February a plant wide power shutdown occurred which turned all systems off. One of the safety interlocks on our vacuum facility assures that the high vac valves are closed off in the event of a power failure. Hence, the vacuum is locked in and no diffusion pump oil can backstream into the chamber prior to the oil cooling off. On Monday, 9 February the system was immediately restarted.

2. The first mandatory shutdown due to thruster malfunction occurred on 27 March. It was observed that the thruster was not firing each time the DI circuit was triggered, indicating a problem with either the ignitor plug or DI circuit. The problem was pinpointed as being the ignitor plug and the chamber was opened to investigate the situation. Visual inspection of the ignitor plug and cathode revealed the existence of a heavy carbon like deposit which filled most of the hole through which the plug discharges to fire the thruster. This deposit was apparently causing blockage of the microplasma produced by the plug discharge so that it could not enter the interelectrode gap. It was thought that the excessive carbon deposit in this region might be the result of the cemented inner propellant rod because the location of the cement seam is very nearly at the center of the hole in the cathode. The only way to prove or disprove this conjecture would be to run an equivalent number of discharges on the system from the same starting

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TABLE I. CHRONOLOGY OF ENDURANCE TEST

	Date of Shutdown	Date of Start Up	Number of Discharges	Reason for Shutdown	Total Impulse Milestone
	–	29 Jan	16,577	Start Test	0 lb-s
1	8 Feb	9 Feb	137,981	Facility	
2	27 Mar	27 Mar	750,097	Mandatory	
3	31 Mar	15 Apr	810,705	Mandatory	
4	17 Apr	20 Apr	837,958	Facility	
5	21 Apr	21 Apr	848,222	Facility	
6	3 May	3 May	1,019,349	Inspection	5,273 lb-s
7	6 May	6 May	1,031,032	Facility	
8	7 May	7 May	1,045,456	Inspection	
9	10 May	25 May	1,070,219	Facility	
10	4 June	6 June	1,273,953	Facility	
11	22 June	22 June	1,447,755	Inspection	
12	25 June	1 July	1,458,249	Mandatory	
13	19 July	19 July	1,742,241	Inspection	
14	30 July	30 July	1,787,343	Inspection	
15	3 August	3 August	1,801,236	Inspection	
16	20 August	20 August	1,996,135	Inspection	10,027 lb-s
17	2 Sept	2 Sept	2,129,129	Inspection	
18	15 Sept	15 Sept	2,206,745	Inspection	
19	16 Sept	16 Sept	2,211,633	Mandatory	
20	4 Oct	–	2,445,785	End Test	12,183 lb-s

conditions and use inner propellant sections which were not cemented together. This being an impractical approach at the time, it was decided to simply clean the carbon deposit out of the hole and resume testing while further consideration of the problem was carried out. This was done and the system put back on test the afternoon of the same day.

3. On the morning of 31 March the thruster was observed to be discharging prior to receiving the firing command and prior to attainment of full voltage on the capacitor bank. This condition is normally attributable to development of a high resistance carbon track somewhere between thruster anode and cathode. The system was shut down for usual inspection of the nozzle area. It was found that one of the outer propellant sections had moved over the fuel retaining shoulder because anode erosion had decreased the shoulder height in the vicinity of that section to less than the amount required to retain it. When this occurred the displaced section bridged the gap between anode and cathode thereby causing the observed premature discharging. The severity of the observed localized anode erosion became a matter of great concern because in all previous development of pulsed plasma systems electrode erosion had not represented a system failure mode. Various ideas on how to eliminate this failure mode have been generated since this first observation and these are discussed in section 6.0 but for purposes of continuing testing it was decided to manufacture a new anode and evaluate an alternate material during the remainder of the endurance test. An anode was machined out of Molybdenum and the thruster placed back on test on 15 April. Because there was no way to tell precisely when premature firing began, the count and total impulse were set back to 810,705 and 4201 lb-s; respectively, which corresponded to those values existing when the most recent thrust measurement was made. The counter actually read approximately 840,000 when the multifiring was first discovered on the morning of 31 March.

4. Plant wide power shutdown on Saturday, 17 April. System restarted Tuesday morning 20 April (Monday was a holiday).

5. A sheet of Mylar used to insulate the thruster from the thrust balance platform was interfering with thrust measurement because the tape holding it in place came undone and the Mylar was hitting against the ball calibrator trough. The chamber was opened and the problem rectified that same day.

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6. The chamber was opened to replace the Molybdenum anode electrode with one made from Tungsten so that this material could be evaluated as well.

7. Power failure occurred overnight. System back on test the next morning.

8. System shut down to inspect the Tungsten anode.

9. Vacuum gage failure caused high pressure cut-off safety interlock to shut down thruster. Gage repaired. Hi Vac system required cleanup and replacement of oil. Testing was discontinued for approximately 2 weeks while the vacuum system was refurbished.

10. Closed loop chilled water system used to cool the diffusion pump shroud and baffles malfunctioned. Thermal switch on diffusion pumps shut system down.

11. Inspection of Tungsten anode.

12. The thruster was again observed to be misfiring. The same situation was found to exist as did on shutdown number 2 and the same corrective action was taken. Having considered the problem in greater detail at the time of this second occurrence, a modified cathode design was subsequently developed to eliminate the problem. It is interesting to note that this second occurrence happened after almost exactly the same number of discharges as the first (i.e., approximately 750,000 the first time and 1,460,000 the second time). The modified cathode design was not incorporated prior to the system being put back on test because time was needed for manufacturing. Details of the new cathode design and other modifications made to eliminate the problem of carbon deposits in this critical area are discussed in section 6.0. A modified Molybdenum anode electrode was also installed for evaluation prior to resuming testing.

13. A low thrust reading led to the suspicion that one or more propellant sections was not feeding properly. Suspicions were confirmed upon inspection of the thruster when it was seen that the negator spring feeding one of the outermost sections was twisted. Prior to initiating the test, these outer feed springs had to be fabricated by spot welding two stock springs together because no single stock size could be procured as a stock item which was long enough. The protruded spot weld location had been tripped by one of the propellant support brackets, causing the spring to twist. Future hardware will incorporate specially made

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springs which are long enough so that spot welding two short springs together can be eliminated.

14. Inspection of the Molybdenum electrode.

15. Thrust balance thought not to be functioning properly. System shut down to check out balance. Recorder imbalance found to be the problem. Recorder fixed and recalibrated.

16. Chamber opened for visual inspection of thruster by AFRPL personnel upon reaching 10,000 lb-s goal.

17. The newly designed cathode was installed with modifications to eliminate carbon deposit within the igniter hole.

18. A newly machined copper anode including design modifications was installed.

19. Misfiring was observed a third time. Roughly 750,000 discharges had been accumulated since the last occurrence indicating that the modified cathode design, although apparently eliminating carbon build-up in the hole itself, was not the only modification required to eliminate the misfiring problem. Further consideration of this problem led to certain other modifications to the plug housing which are discussed in section 6.0.

20. Endurance test voluntarily ended since the 10,000 lb-sec goal had been exceeded and it was necessary to begin thruster refurbishment (see section 6.0).

5.0 ISOLATED PROBLEM AREAS

Two fundamental problems were encountered during the endurance test run on this system. These were anode electrode erosion and carbon deposition on the ignitor plug. Both of these problems, having necessitated mandatory shutdown of the thruster during testing, represent possible system failure modes. Modifications were made to the thruster during the endurance test in attempting to resolve these problems and although testing allowed for some partial evaluation of those modifications, the duration of the testing did not allow for a definite assessment of the modifications.

Regarding electrode erosion it is evident that no problem exists with the thruster cathode. The original thruster cathode, fabricated from copper, was used successfully throughout the test with the exception of the last 115,000 discharges when a modified

cathode design was incorporated in attempting to eliminate carbon build-up within the ignitor plug cavity. Erosion of the thruster anode could be considered severe enough to represent a potential failure mode for the system because of the resulting reduction in height of the propellant retaining shoulder in the vicinity of the outermost (i.e., furthest downstream) propellant section. It is interesting to note that the majority of the anode surface exposed to the discharge remains practically erosion free. Erosion begins to become visually apparent at the downstream end of the electrode and the eroded area very slowly propagates upstream as discharges accumulate. An engineering drawing of the anode electrode as initially manufactured for this test is presented in Figure 5. A qualitative picture of the erosion pattern as it has been observed to progress during testing is presented in Figure 6 to illustrate the above statements. The area of severe erosion begins to enter the region between propellant rods after approximately 6 - 700,000 discharges and continues to propagate into this region as time goes on. A photograph of one of the electrodes used during the test is presented in Figure 7 showing the degree of erosion after approximately 820,000 discharges. Anode electrodes were manufactured from copper, tungsten, and Molybdenum during this program to assess the relative erosion resistance of each under actual long-term thruster operating conditions. Quantitative data on the degree of erosion was not generated for each of these materials but visual observations at various times during testing were made. On the basis of these observations it can be said that the general character of the erosion pattern depicted in Figure 6 was the same regardless of which material was used and that thruster propulsive performance remained unchanged with material substitution. Indications were that Molybdenum would last longer than either copper or tungsten. Presently it is questionable whether any of these materials would survive the required life for the system with the original geometric configuration that was tested and as a result a program will be carried out to investigate the problem of erosion as it relates specifically to this thruster system. A certain amount of basic independent research will be performed during that program but the major portion of the work will be an experimental evaluation of several different approaches to elimination of anode erosion of the fuel retaining shoulder as a potential failure mode for this system over its design life. The electrode erosion program is scheduled to commence in the immediate future. The prognosis for developing a solution to this problem is excellent at this time because of several ideas of significant merit.
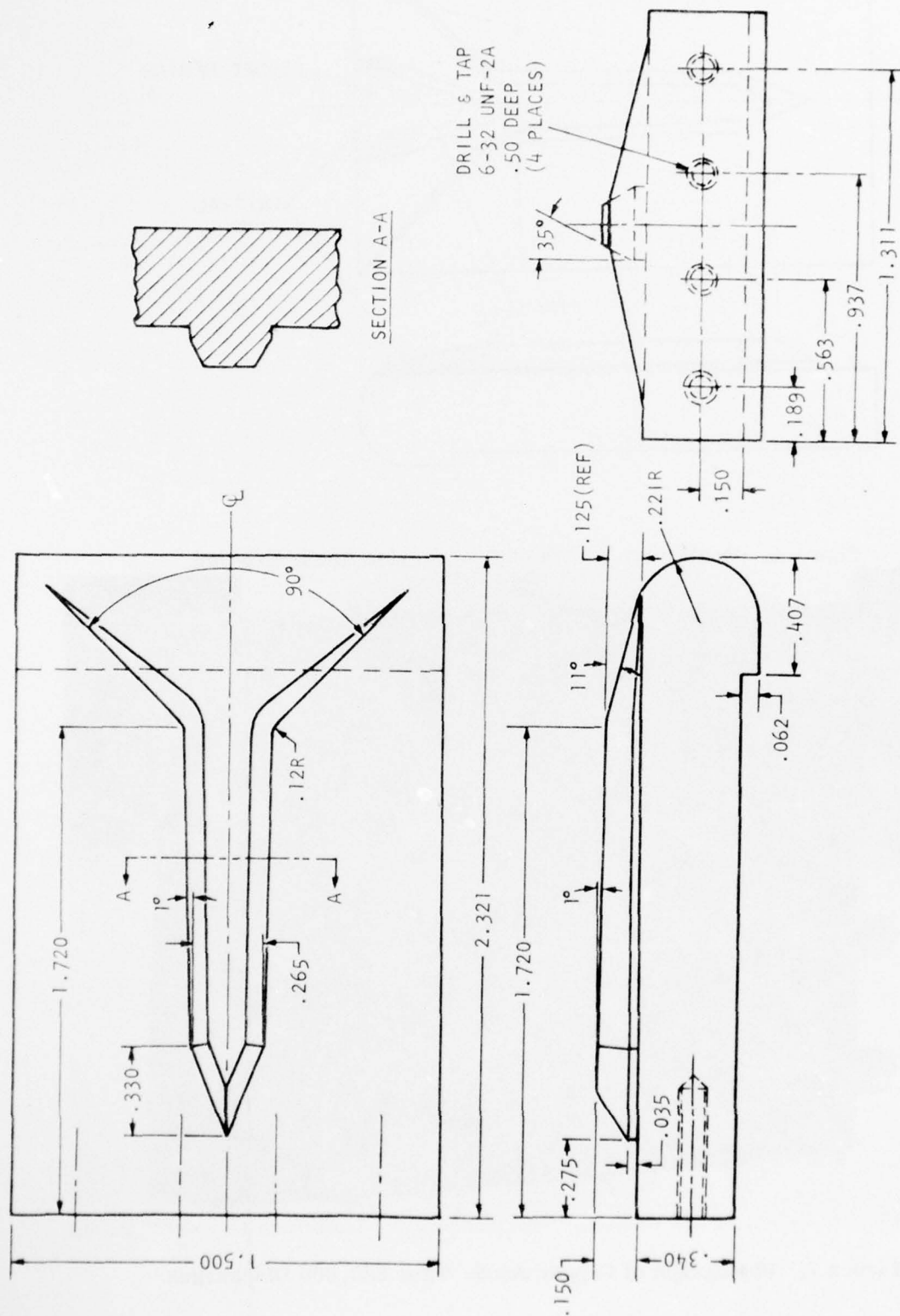
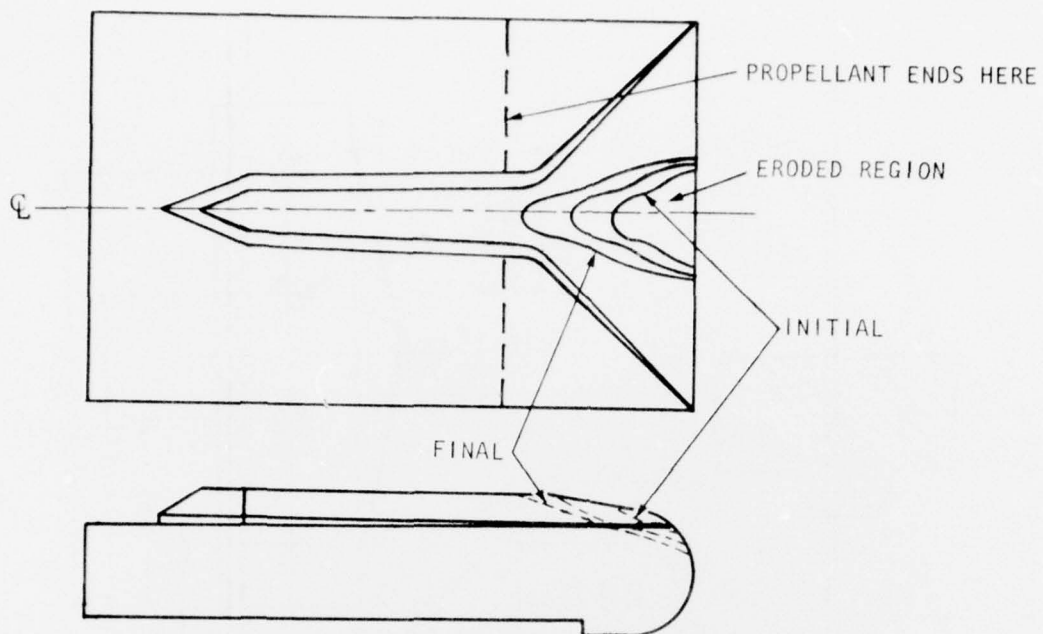Figure 5. Original Anode Design

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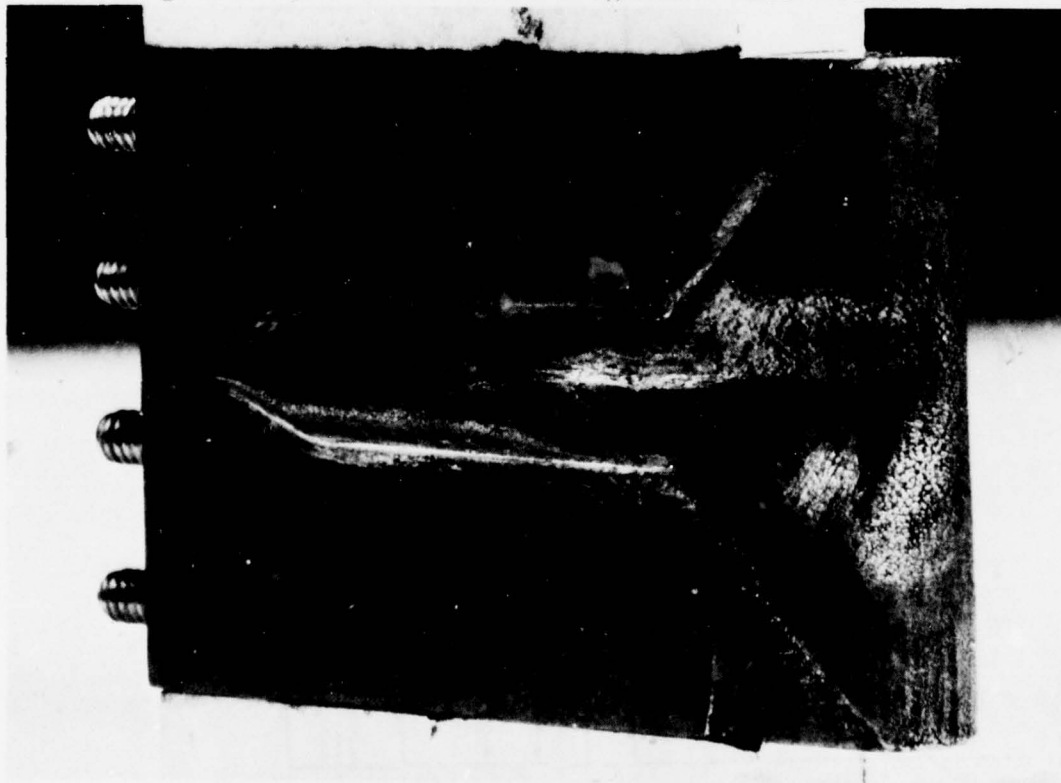Figure 6. Qualitative Sketch of Progressive Anode Erosion

Figure 7. Photograph of Copper Anode After 820,000 Discharges

The thruster misfiring problem was at first thought to be a result of carbon deposition in the hole through which the plug microplasma must pass. Upon changing the cathode geometry in the vicinity of this hole (see section 6.0 for details) and running sufficiently long to evaluate this change it became apparent that the problems encountered were only partly a result of carbon build-up around the downstream periphery of the hole. It was determined that the deposition of carbon on the surface of the ignitor plug itself, causing a short (about 10 ohms) to develop between plug anode and cathode, was accountable for not allowing the plug to break down upon application of the firing pulse. Further analysis of the problem led to a modification in the design of the ceramic plug housing. This modified housing was not installed on the thruster during the endurance test because the need for the modification did not become apparent until very nearly the end of the test. The modified housing will be installed prior to delivery of the thruster to the AFRPL for their in-house testing and evaluation program. Cathode and ignitor plug housing modifications are covered in section 6.0. Presently it is not yet known whether or not this abnormal growth in the carbon deposit could in part be due to the carbonaceous layer of cement that was used between the bonded inner propellant rod (see section 2.4). Future testing with a solid inner propellant rod will hopefully clarify the question.

In addition to the two fundamental problem areas noted in the preceding paragraphs, some minor functional problems arose primarily because of the lack of prior experience with a helically coiled propellant system such as this one. In particular the rigidity of the propellant support structure and the importance of the minor deviations from planar geometry were underestimated when the system was first designed. In addition, the frictional forces between propellant rods were also underestimated primarily for the same reason. The consequent large spring forces necessary to feed the propellant rods led to structural deflections which had a tendency to load up the support structure and occasionally bind or misalign the propellant rods. The experience gained from working with the system allowed for formulation of design modifications which in effect eliminated the need for such large springloads and which yielded a more structurally rigid assembly. These modifications are discussed in the next section of this report.

Upon completion of the endurance test, it was noted that a very small quantity of liquid impregnant had leaked out of the end cap seal of the anode (center) stud of one of the KF-film capacitors. The minute leakage was observed under one of the phenolic

insulating caps which screws onto the stud. During fabrication of the capacitors (see Ref. 5), a redundant sealing cap was to be provided over the seal as a separate operation. To simplify assembly procedures this redundant cap was soldered in place at the same time the capacitor core was soldered to the stud. Thus, seal redundancy was not in fact realized. Either the simple two-step procedure should be used in the future, or a minor redesign of the seal should be carried out. In any case, it would be possible to epoxy the phenolic cap onto the stud to insure a good seal at that point.

6.0 CORRECTIVE DESIGN MODIFICATIONS

As a result of the testing that was done with this thruster, certain design modifications were proposed which would lead to a more reliable overall system capable of running for a longer period of hands-off operation. These modifications included

(i) Increasing structural rigidity and minimizing frictional loads on propellant sections to minimize feed spring forces.

(ii) Having feed springs of proper length manufactured to custom fit the thruster requirements so that welding separate springs together could be eliminated.

(iii) Modification of dissipative component heat sinking within the power converter to insure proper output power under the most severe thermal environment in vacuum.

(iv) Modification of ignitor plug housing to incorporate changes in the original design that would eliminate the build-up of carbon deposits on the surface of the plug.

(v) Fabrication of a new anode from Molybdenum which would incorporate changes to the propellant retaining shoulder leading to longer testing time before replacement, if necessary.

(vi) Fabricate new Teflon propellant rods having full total impulse capability and eliminating the cemented inner section by manufacturing the inner sections as one piece.

(vii) Redesigned spring feed system to eliminate the heavier spring motor and pulley assembly used during the endurance test.

The above modifications were suggested because of the desire to perform a second endurance test on the thruster system for generation of the entire total impulse capability. This test will be performed in-house by the AFRPL and will have as its

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objectives the demonstration of full capacitor and power converter life and functional reliability of the propellant feed system to feed the entire amount of required propellant. In addition, this test is to serve as an evaluation of design modifications made on the ignitor plug housing, cathode, and anode retaining shoulder and the propellant feed system. The following text describes the modifications that were subsequently made to the system.

Structural rigidity of the propellant support assembly was enhanced by redesigning the configuration of the center support and alignment ring. In addition, this main structural member was machined out of a continuous aluminum plate whereas it had previously been manufactured from NEMA G-10 phenolic. The propellant support brackets were also redesigned and machined from .25 inch (6.35 mm) aluminum as opposed to the original .125 inch (3.68 mm) aluminum plate. The brackets closest to the nozzle entrance on either side were fitted with a spring loaded Teflon roller to insure that the propellant rods would be pushed down onto the anode surface just prior to coming in contact with the fuel retaining shoulder while maintaining free motion of each propellant section under thermally expanded conditions. A photograph of this arrangement is presented in Figure 8.

In order to minimize the spring force required to feed the individual propellant sections, the idea of separating the sections using a thin aluminum rail around the periphery of the helix was expanded upon. It was decided to separate the middle from the outer sections of propellant as well as the inner from the middle sections. It was also seen (as a result of the experience generated from working with the system) that the sections should be separated on both the top and the bottom because of the tendency for them to tilt under spring loading and possibly bind together if separated on only one side. Thus, "splitter rails" which separate the three propellant sections from one another on both the top and bottom were incorporated into the refurbished system. The splitter rails were specially coated with TUFRAM-4 Teflon coating process to minimize friction between the rails and the propellant sections. In addition, a continuous aluminum skin was fabricated which ties together all of the propellant support brackets around the outside circumference. A photograph of the complete refurbished propellant support structure is presented in Figure 9.

The method whereby the propellant sections are fed was also changed on the refurbished system. Each section is now provided with a separate set of springs positioned between them in the .040 inch (1.02 mm) space produced by the splitter rails.
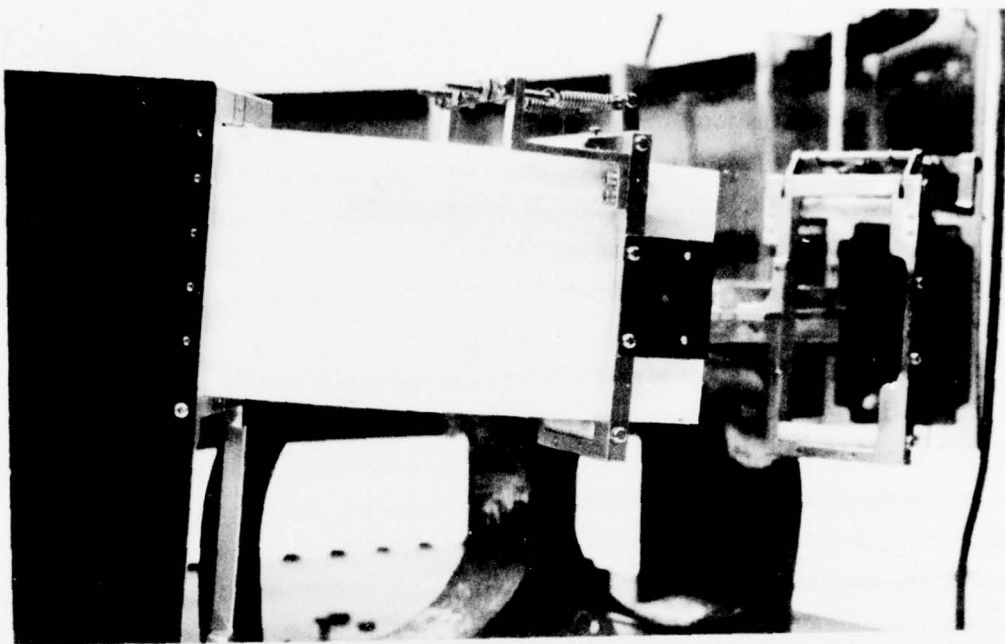
Figure 8. Spring Loaded Roller Assembly

Figure 9. Refurbished Propellant Support Structure

The force required to move the propellant rods has been reduced by a factor of ten in the case of the innermost section and to a comparable degree on the other two sections as a result of these modifications. Special order negator springs having appropriate dimensions and force were manufactured for the refurbished system by Ametek Hunter Spring Division, thereby eliminating welding of individual springs to obtain one having the required length.

The breadboard power converter was returned to Wilmore Electronics for repackaging with the objective of properly heat sinking those components which generate the most heat. This task was successfully accomplished by Wilmore, as demonstrated during subsequent thermal vacuum testing of the completely refurbished thruster system. This data is presented later in this report.

Build-up of carbon deposits on the surface of the ignitor plug can have the ultimate effect of causing the plug to misfire. This is so because once the exposed surface of the semiconductor becomes completely coated with carbon the path of least resistance is directly through the carbon layer and not through the first monolayer of semiconducting material as it should be for proper firing to occur. Microthrusters, having typical discharge energies up to 20J, are constructed with the ignitor plug press fitted into the thruster cathode. The plug cathode (outer cylinder) and thruster cathode are therefore at essentially the same potential. This technique is not appropriate at the larger discharge energies characteristic of millipound level thrusters because it has been observed that plug cathode erosion becomes very severe at the higher energies. The approach used on early millipound test thrusters was to insulate the plug cathode from the thruster cathode by casting a thick layer of epoxy around the plug cathode. A one ohm resistor was then connected between plug cathode and thruster cathode to insure a high impedance path back to ground through the plug with respect to the path back to ground through the thruster cathode. That technique eliminated the problem of plug erosion, and although it was observed that thruster cathode erosion in the vicinity of the plug became more pronounced, the level of erosion was still tolerable. Experimentation with thrusters utilizing still higher discharge energies forced the abandonment of that method as well because the degree of cathode erosion near the plug was becoming unacceptable, eventually leading to destruction of the epoxy insulation surrounding the plug. It was during later experimentation with a conical thruster nozzle that the idea of placing the ignitor plug in an insulated housing in the immediate vicinity of the

desired ignition location was developed. This technique worked well, essentially eliminating erosion of both the plug cathode and the thruster cathode. The first long term thruster endurance test run using this technique was performed for NASA Langley R.C. (see Ref. 7). During that test approximately 2.7 million discharges were accumulated at 100% duty cycle with no test interruptions. A total impulse of 3100 pound-seconds (13,790 Ns) was generated and 1.74 lb (.79 kg) of Teflon propellant were consumed. In the wake of such a result it seems surprising that ignitor plug misfiring was encountered during the current endurance test after only 750,000 discharges. When one considers this result on the basis of total propellant mass consumed, and it is noted that approximately 2.4 lbs (1.09 kg) of propellant were used during the 750,000 discharges prior to plug misfiring during the current test, it is understandable. However, the carbonaceous layer of cement between the inner propellant rod may have also contributed to the problem.

The first step taken in attempting to resolve the misfiring problem was a redesign of the cathode surface in the vicinity of the ignition hole. The original design called for a 0.25 inch (6.35 mm) dia. chamfered hole in the flat cathode surface as shown in Figure 10a. Carbon deposition on the downstream lip of this hole and subsequent buildup of deposits there caused the hole to become almost entirely closed off by the time the first misfiring was observed. It was reasoned at that time that the downstream lip of the hole represented a stagnation region to the accelerating plasma and that something should therefore be done to minimize the frontal area of this stagnation region and also try to deflect the flow of plasma away from the hole itself. Hence, the configuration shown in Figure 10b was arrived at. Note that the .25 inch diameter circular hole was changed to a rectangular hole 0.25 inches long by 0.375 inches (9.53 mm) wide. The surface of the cathode upstream of this hole is slanted downward at a 5° angle to deflect the plasma flow away from the hole. The cross section of the hole on the downstream side is decreased in area by beginning a constant slope of 5° to the full thickness of the electrode from an initial thickness of .01 inches (.254 mm). The modified cathode was not installed on the thruster immediately following the first misfiring occurrence because the manufacturing time would have caused the system to be down for a longer period of time than was desired. Hence, the original configuration was replaced on test after deposits were cleaned out of the hole and off the surface of the plug.
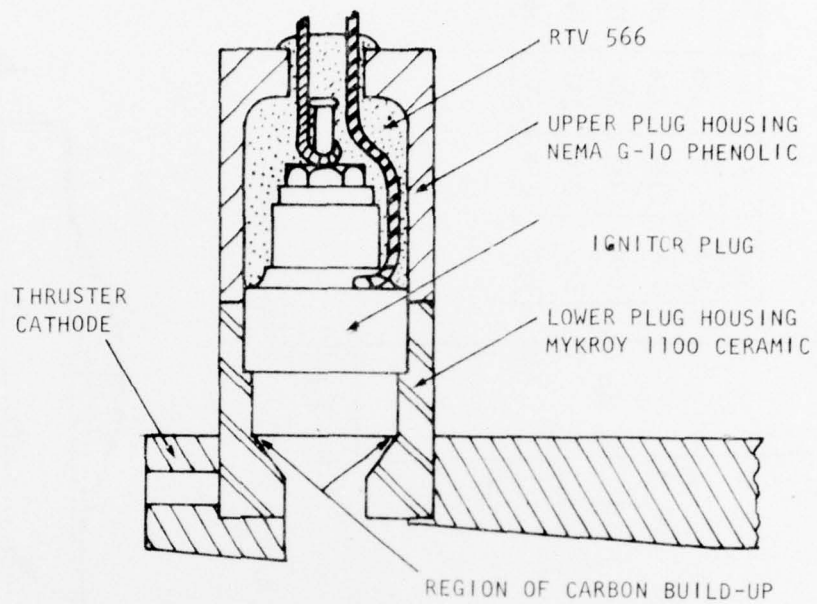
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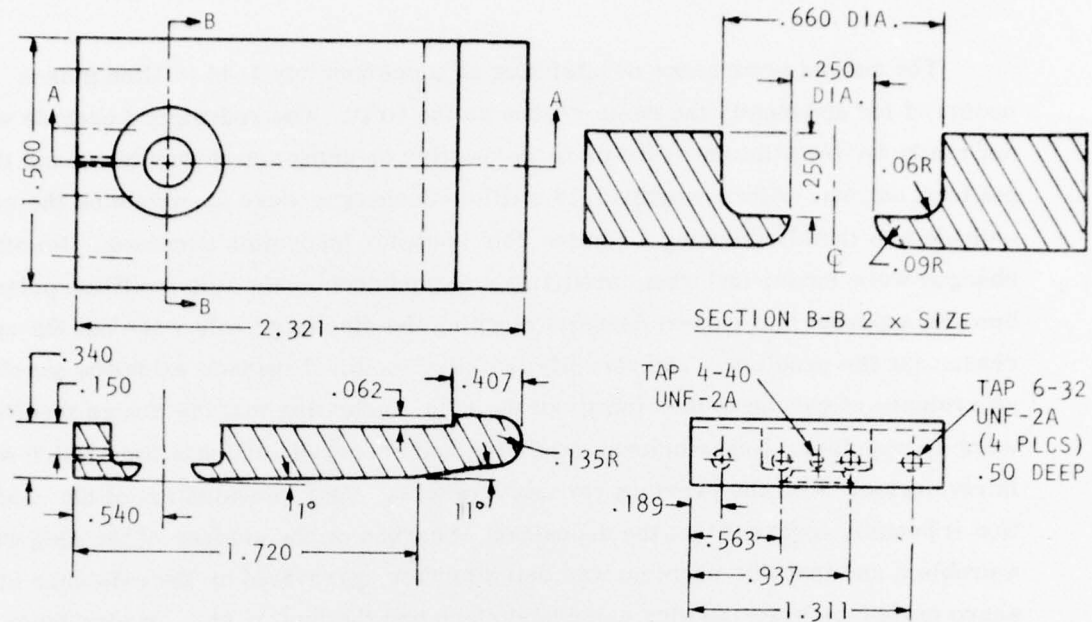
(a) Original Design
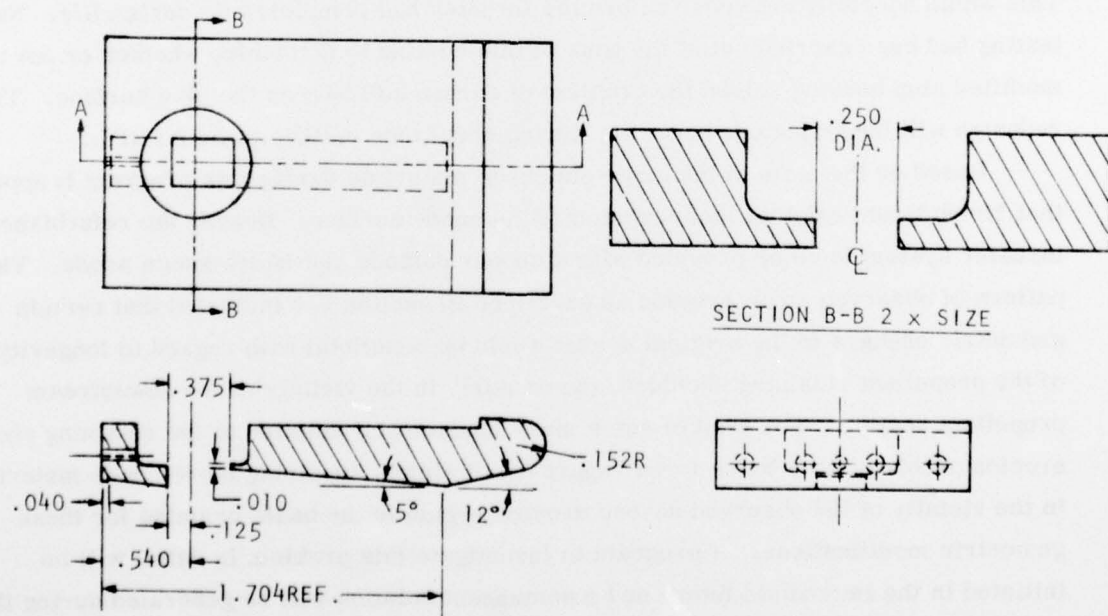
(b) Modified Design

Figure 10. Modifications to Cathode Design in the Vicinity of the Ignition Hole
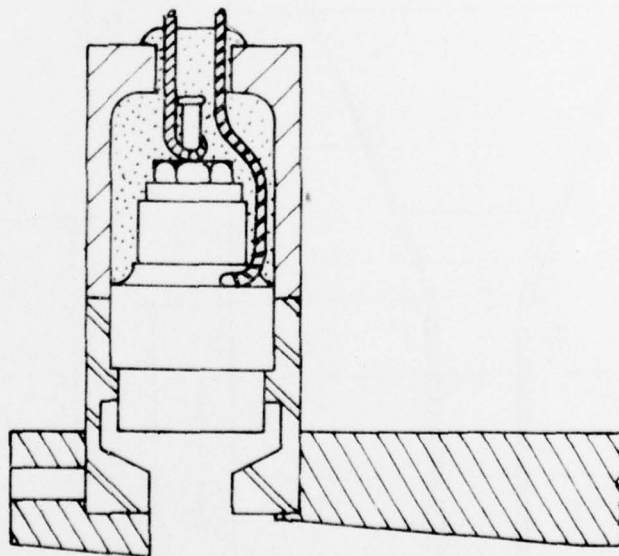
The second occurrence of misfiring at approximately 1.46 million pulses occurred for apparently the same reason as the first. The redesigned cathode was not ready for installation so the same corrective cleaning action was taken and the system put back on test. After roughly 2.13 million discharges were accumulated the new cathode was installed during a routine shut down for inspection purposes. No other changes were made, and when misfiring occurred once again at 2.2 million pulses it became apparent that carbon deposits clogging the discharge hole were not the only reason for the problem. The recently installed modified cathode exhibited absolutely no evidence of carbon in the vicinity of the hole, indicating that the design modifications were appropriate. Nevertheless, misfiring had occurred, and at a time which was fairly periodic with the previous two occurrences. Upon reevaluation of the configuration it became apparent that the deposition of carbon on the surface of the plug was still a problem and that this problem was being further aggravated by the existence of a sharp corner between the plug cathode surface and the ignitor plug housing inner walls. The internal configuration of the ceramic plug housing was therefore redesigned as shown in Figure 11.

The space provided around the circumference of the plug cathode, between the plug and the ceramic housing would now prevent accumulation of carbon on the surface of the plug cathode until the surrounding cavity was completely filled with carbon. This would hopefully not occur before the thruster had completed its design life. No testing had been carried out at the time of this writing to determine whether or not the modified plug housing solved the problem of carbon build-up on the plug surface. The redesign will be evaluated, however, during endurance testing at the AFRPL.

Based on the tests of the three electrode materials during this program it appears that Molybdenum exhibits least erosion of the anode surface. Hence, the refurbished thruster system is being provided with a copper cathode and Molybdenum anode. The pattern of observed anode erosion as explained in section 5.0 indicated that certain geometric changes to the original design would be beneficial with regard to longevity of the propellant retaining shoulder, particularly in the vicinity of the downstream propellant section. For want of some more fundamental solution to the retaining shoulder erosion problem, the "brute force" approach of simply supplying more anode material in the vicinity of the observed severe erosion region is the basic premise for these geometric modifications. A program to investigate this problem in detail will be initiated in the immediate future and a permanent solution will be generated during that program.

RTV 566

UPPER PLUG HOUSING
NEMA G-10 PHENOLIC

IGNITOR PLUG

LOWER PLUG HOUSING
MYKROY 1100 CERAMIC

THRUSTER
CATHODE

REGION OF CARBON BUILD-UP

A. ORIGINAL HOUSING DESIGN

B. MODIFIED HOUSING DESIGN

Figure 11. Modifications to Ignitor Plug Housing

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SECT.	X	Y	Z
A-A	.342	.338	- -
B-B	.344	.336	.265
C-C	.346	.334	.276
D-D	.348	.332	.288
E-E	.350	.330	.300

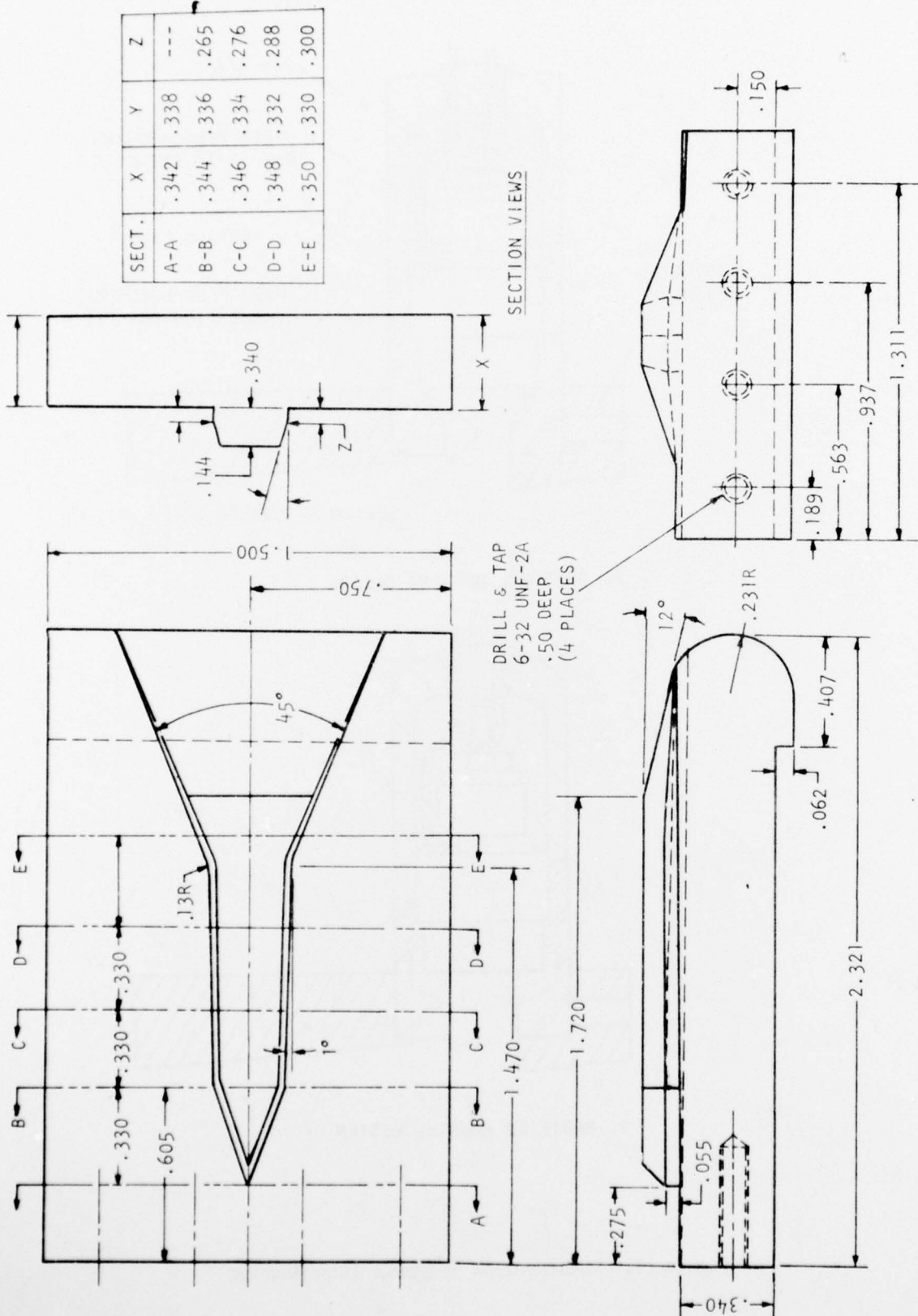SECTION VIEWS

DRILL & TAP
6-32 UNF-2A
.50 DEEP
(4 PLACES)

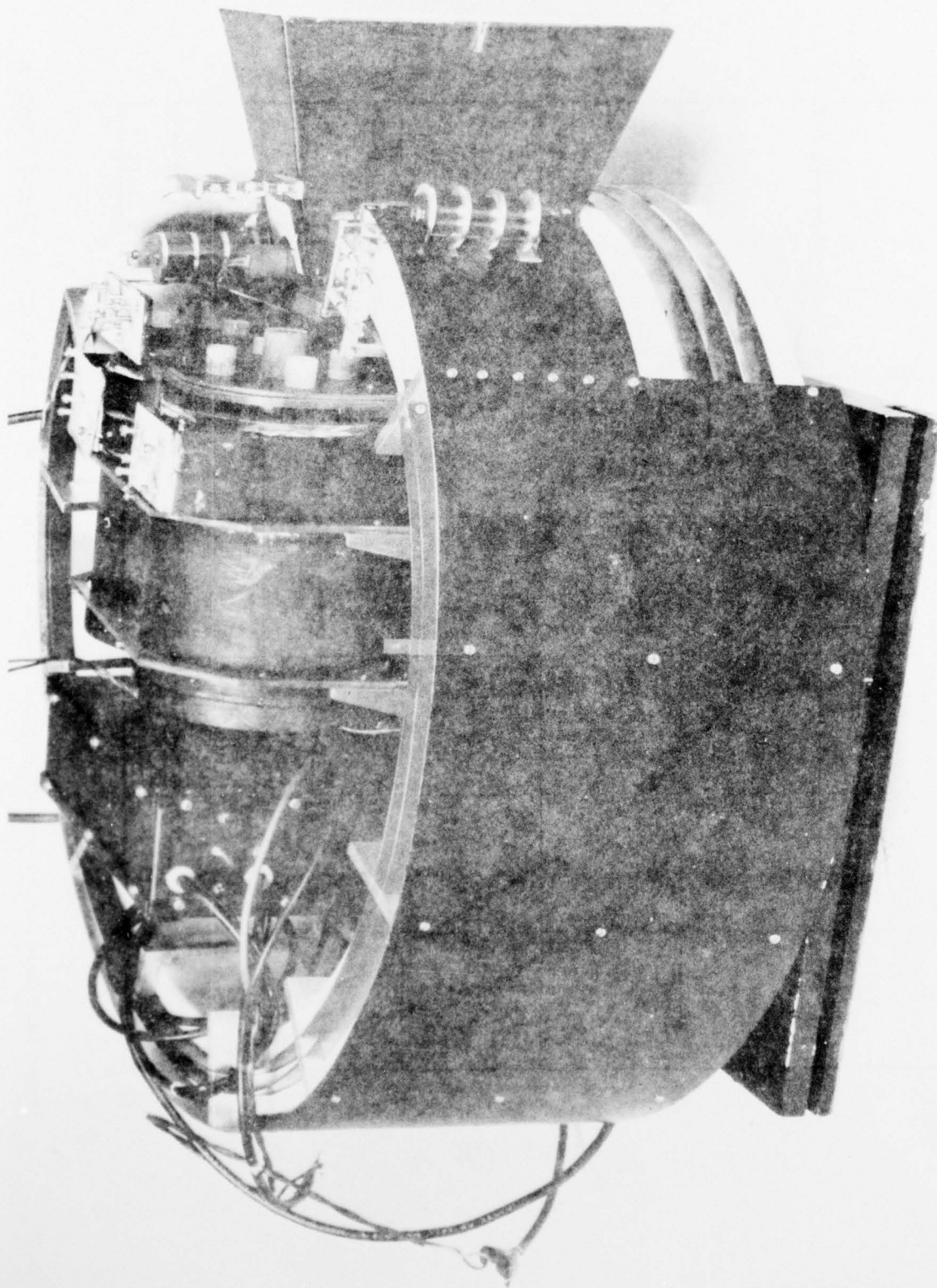Figure 12. Modified Anode Design

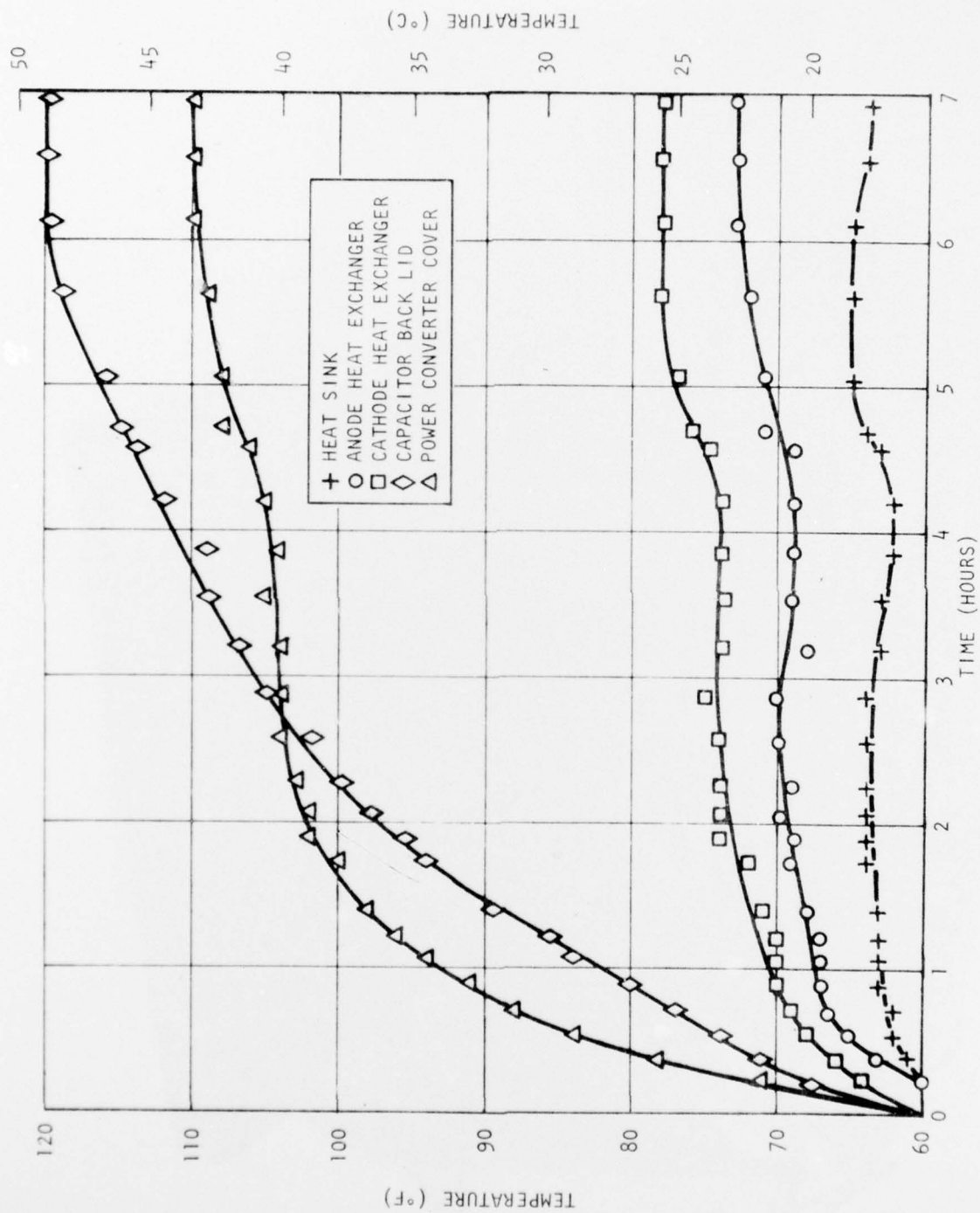Figure 13. Refurbished Thruster System

Figure 14. Temperature as a Function of Time at Various Locations on Refurbished Thruster System

Figure 12 is a reduced engineering drawing of the modified anode design. Changes from the original include an increase in shoulder height at the downstream end obtained by eliminating the 1° downward slope of the top surface; an increase in the distance to which the flared end penetrates between the propellant rods; an increase in the height of the flat portion of the shoulder from .032 inches (.81 mm) to .055 inches (1.4 mm); and a decrease in the chamfer angle (between the top of the flat shoulder and the top surface of the electrode). All of these modifications were incorporated in order to provide a larger amount of electrode mass on the downstream end of the shoulder, thereby allowing for more material to erode away before the shoulder height becomes reduced to a point where it can no longer retain the spring loaded propellant rod.

Photographs of the completed refurbished thruster system are presented in Figure 13. Thermal vacuum testing of the complete system (including the refurbished power converter) was performed at Fairchild Republic and the results, presented in Figure 14, indicate completely compatible operation in vacuum.

7.0 CONCLUSIONS

This program was the first attempt at the design, assembly and testing of a completely integrated millipound thrust level pulsed plasma thruster system incorporating the advanced high performance side-fed nozzle configuration, helical propellant rods, increased state-of-art high energy density capacitors and a packaged breadboard power converter. Many questions which existed at the onset of this program have been answered and some new questions and problems have surfaced which have not been previously encountered. Some of the problems encountered were solved while others will receive more attention during the course of development, but the prognosis for generating solutions to the problems which still exist is excellent. When one considers the fact that this propulsion system exceeds by a factor of ten the amount of thrust delivered, and by a factor of 40 the amount of total impulse capability, while at the same time having considerably higher efficiency than any previously assembled pulsed plasma propulsion system, one concludes that a major jump in the state-of-art has been achieved.

The major achievements of this program were;

(i)   Demonstration of the capability to feed separate solid propellant sections from the sides of the discharge channel so that high performance could be realized and maintained.

(ii)   Demonstration of the principle of storing large amounts of solid propellant using helical propellant rods to minimize volumetric requirements.

(iii)   Demonstration of the long term vacuum compatibility and life of high energy density KF-film capacitors.

(iv)   Demonstration of the thermal vacuum compatibility of the complete thruster system, including power converter.

(v)   Demonstration of the long term retention of high performance repeatable thrusting capability of the system.

(vi)   Demonstration of more than 12,000 lb-sec of total impulse during an endurance vacuum test at an I_{s_p} exceeding 1600s.

Future testing at the AFRPL will be performed for the 37,500 pound seconds capability of the system. The same four capacitors as were used during the endurance test performed at Fairchild Republic will be used during testing at the AFRPL. Hence, about 30% more life than is required will be demonstrated on these units at the completion of testing at the AFRPL. The modified cathode, anode, and ignitor plug housing designs will be evaluated during that test. Electrode erosion will be studied in-depth at Fairchild Republic in another program to eliminate the potential thruster failure mode resulting from excessive fuel retaining shoulder erosion. Although investigations with respect to determination of material and/or geometric configuration effects on erosion level will be carried out, the ultimate objective will be elimination of this potential failure mode by whatever means possible. Several noteworthy concepts concerning this problem have already been formulated and a high degree of confidence in being able to solve the problem is therefore prevalent.

Future work on this system will be directed at bringing the system up to the status of fully flight qualified hardware.

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